TECHNICAL NOTES

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 625

SPINNING CHARACTERISTICS OF WINGS

IV - CHANGES IN STAGGER OF RECTANGULAR CLARK Y BIPLANE CELLULES

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SUMMARY

Rectangular Clark Y biplane cellules having zero and -0.25 stagger, the gap equal to the chord, and 0° decalage were tested on the N.A.C.A. spinning balance in the 5-foot vertical tunnel. The aerodynamic characteristics of the models and a prediction of the angles of sideslip for steady spins for airplanes using these wing arrangements are given. There is included an estimation of the yawing moment that must be furnished by parts of the airplane other than the wing to balance the inertia couples and wing yawing moments for spinning equilibrium. The effects on the spin of changes in stagger and of variation in some of the important parameters are discussed and the results are compared with those for a similar biplane cellule with 0.25 positive stagger tested earlier.

It is concluded that, with the values of stagger considered, for a conventional biplane having equal upper and lower rectangular Clark Y wings, gap equal to the chord, and zero decalage: The airplane will generally spin with inward sideslip, which, in some cases, may exceed 20°; for angles of attack through 50°, the sideslip generally will become more inward as the stagger becomes more negative and, for an angle of attack of 70° and sometimes of 60°, the inward sideslip will become less as the stagger becomes more negative; the value of stagger for the best spinning characteristics will vary with different types of airplanes; the provision of a yawing-moment coefficient of -0.025 (i.e., opposing the spin) by the tail, fuselage, and interference effects will prevent equilibrium in a steady spin for the values of stagger tested and with any of the parameters used in the analysis; and too much reliance should not be placed on tail arrangement for preventing bad spinning characteristics.

INTRODUCTION

In order to provide necessary aerodynamic data for estimating airplane spinning characteristics from the design features, the N.A.C.A. is conducting an investigation to determine the aerodynamic characteristics of airplane models and parts of airplane models in spinning attitudes.

The portion of the investigation to determine the spinning characteristics of wings, for which the N.A.C.A. spinning balance is being used, includes a study of the effects of variations in airfoil section, plan form, and tip shape of monoplane wings and variations in stagger for biplane cellules. The first and third series of tests reported were made of Clark Y monoplane wings with rectangular plan forms, with square and rounded tips, and with a 5:2 tapered wing having rounded tips (references 1 and 2). The second series, made of a rectangular Clark Y biplane cellule with 0.25 stagger, is reported in reference 3.

The present report is a continuation of reference 3 and gives the aerodynamic characteristics, in spinning attitudes, of a rectangular Clark Y biplane cellule with the gap equal to the chord, zero decalage, and with zero and -0.25 stagger. Also included are comparisons with the cellule having 0.25 positive stagger.

APPARATUS AND MODELS

The tests were made on the spinning balance in the N.A.C.A. 5-foot vertical wind tunnel. The tunnel is described in reference 4 and the six-component spinning balance in reference 5.

The Clark Y wings were made of laminated mahogany with balsa insets for lightness. The span of each wing is 30 inches and the chord is 5 inches. These wings had been used for the tests in reference 3; the only change in the cellule was new strut bracing to give the desired amounts of stagger. Figure 1 is a sketch of the model showing the bracing, balance attachment, and stagger. Figure 2 shows the model (-0.25 stagger) mounted on the spinning balance.

TESTS

In order to cover the probable spinning range, tests were made at angles of attack of 30°, 40°, 50°, 60°, and 70°. At each angle of attack tests were made with values of $\Omega b/2V$ of 0.25, 0.50, 0.75, and 1.00. At each angle of attack and at each value of $\Omega b/2V$ tests were made at sideslip angles of -5°, 0°, 5°, 10°, and 15° for the cellule with zero stagger, and at 0° (α = 70° only), 5°, 10°, 15°, and 20° sideslip for the cellule with -0.25 stagger. The angles of attack and of sideslip were measured in the plane of symmetry at the quarter-chord point of the upper wing, which was also the center of rotation for all tests. Because of variations in individual balance readings, at least one repeat test was made for each condition and an average of the individual measurements was used to compute the coefficients.

The tunnel air speed was 70 feet per second for tests with $\frac{\Omega b}{2V}$ = 0.25 and 0.50, 56 for $\frac{\Omega b}{2V}$ = 0.75, and 42 for $\frac{\Omega b}{2V}$ = 1.00. The Reynolds Number was about 180,000 for the highest air speed and about 137,000 for the lowest. Previous tests showed no appreciable change in scale effects for this range.

RESULTS AND DISCUSSION

The data were converted to coefficient form by means of the following relations:

$$C_X = \frac{X}{qS} \qquad C_Y = \frac{Y}{qS} \qquad C_Z = \frac{Z}{qS}$$

$$C_l = \frac{L}{qbS} \qquad C_m = \frac{M}{qbS} \qquad C_n = \frac{N}{qbS}$$

All coefficients are standard N.A.C.A. form except C_m, which is based on the span rather than on the chord and may be converted to the standard N.A.C.A. form by multiplying by 6. All coefficients are given with the conventional signs for right spins (reference 1).

The coefficients of longitudinal force in the earth system of axes $C_{X''}$ and of all six components of the forces and moments in the body system of axes are given in tables I and II. Sample curves of $C_{X''}$, C_l, C_m, and C_n are given in figures 3 to 6.

The data and attitudes are given for the quarter-chord point on the lower surface of the upper wing at zero radius. The coefficients in body axes may be converted to any other point of rotation in the plane of symmetry by the following relations. The converted coefficients are indicated by the subscript 1.

$$C_{X_1} = C_X \left(\frac{V}{V_1}\right)^2 \qquad C_{Y_1} = C_Y \left(\frac{V}{V_1}\right)^2 \qquad C_{Z_1} = C_Z \left(\frac{V}{V_1}\right)^2$$

$$C_{l_1} = \left[C_l + \frac{z}{b}\, C_Y\right] \left(\frac{V}{V_1}\right)^2$$

$$C_{m_1} = \left[C_m - \frac{z}{b}\, C_X + \frac{x}{b}\, C_Z\right] \left(\frac{V}{V_1}\right)^2$$

and

$$C_{n_1} = \left[C_n - \frac{x}{b}\, C_Y\right] \left(\frac{V}{V_1}\right)^2$$

where x is the distance forward (positive) of the new center of rotation from the quarter-chord of the upper wing.

z, the distance of the new center of rotation below (positive) the lower surface of the upper wing.

b, the span of the wing.

$$\frac{V_1}{V} = \sqrt{\frac{u_1^2}{V^2} + \frac{v_1^2}{V^2} + \frac{w_1^2}{V^2}}$$

$$\frac{u_1}{V} = \cos\alpha\,\cos\beta + \frac{2zq}{b\,\Omega}\left(\frac{\Omega b}{2V}\right)$$

$$\frac{v_1}{V} = \sin\beta + \frac{2xr}{b\,\Omega}\left(\frac{\Omega b}{2V}\right) - \frac{2zp}{b\,\Omega}\left(\frac{\Omega b}{2V}\right)$$

$$\frac{w_1}{V} = \sin\alpha\,\cos\beta - \frac{2xq}{b\,\Omega}\left(\frac{\Omega b}{2V}\right)$$

$$\frac{p}{\Omega} = \cos\alpha\cos\beta$$

$$\frac{q}{\Omega} = \sin\beta$$

$$\frac{r}{\Omega} = \cos\beta\sin\alpha$$

Thus $\alpha_1 = \tan^{-1}\dfrac{w_1}{u_1}$

$$\beta_1 = \sin^{-1}\frac{v_1}{V_1}$$

An analysis was made with the data converted to the quarter-chord point midway between the wings of the biplane (reference 3). The analysis showed that the sideslip required was generally about 2° less than it was for the original data. In other details the variations were quite similar.

The data are believed to be correct to within the following limits:

$C_{X''}$, ±0.02; C_X, ±0.02; C_Y, ±0.01; C_Z, ±0.02;

C_l, ±0.001; C_m, ±0.002; C_n, ±0.001

No corrections have been made for the effects of jet boundaries, scale, or interference of the balance, struts, or bracing system.

Generally, $C_{X''}$ decreases as the stagger decreases (fig. 3). This result may normally be expected because of the blanketing of the upper wing by the lower wing. The variation of C_l with stagger changes sign with increase of angle of attack (fig. 4). The values of C_l at 30° angle of attack are more positive for the negative stagger, and at 70° angle of attack are more positive for the positive stagger. The changes in C_l with $\Omega b/2V$ are irregular. The values of C_m increase as the stagger decreases (fig. 5). Part of this increase is due to measuring the moments about a fixed point on the upper wing so that de-

creasing the stagger means moving the lower wing forward with respect to this fixed point. (See fig. 1.) The values of C_n are small and show no general tendency to change with the stagger (fig. 6).

ANALYSIS

An analysis of the data was made to show the effects of certain parameters on the steady spinning characteristics of an airplane using these types of biplane cellule. The method of analysis with the assumptions used and the errors involved is given in reference 1. For convenience the method of analysis is briefly described.

Formulas used in the analysis.-

$$\frac{\Omega b}{2V} = \sqrt{\frac{-C_m}{3.84\ \mu \sin 2\alpha} \times \frac{b^2}{{k_Z}^2 - {k_X}^2}} \tag{1}$$

$$C_l = C_L\left(\frac{{k_Z}^2 - {k_Y}^2}{b\sqrt{{k_Z}^2 - {k_X}^2}}\right)\sqrt{\frac{-C_m \tan \alpha}{2\mu}}$$

$$+ 1.02\left(\frac{{k_Z}^2 - {k_Y}^2}{{k_Z}^2 - {k_X}^2}\right)\left(\frac{-C_m \sin \beta}{\cos \alpha}\right) \tag{2}$$

$$C_n = C_l \cot \alpha\left(\frac{{k_Y}^2 - {k_X}^2}{{k_Z}^2 - {k_Y}^2}\right) \tag{3}$$

Parameters.- Because the wing loading, aspect ratio, radii of gyration, and pitching moments are mostly dependent upon the characteristics of the particular airplane, values of these variables covering the range for normal biplanes have been used in the analysis. A mean of these values was chosen that gave the following parameters:

Relative density of the airplane to air

$$\left(\mu = \frac{W}{g\rho bS} = \frac{m}{\rho bS}\right)\ \mu = 5$$

Pitching-moment inertia parameter, $\dfrac{b^2}{{k_Z}^2 - {k_X}^2} = 80$

Rolling-moment and yawing-moment inertia parameter,

$$\frac{{k_Z}^2 - {k_Y}^2}{{k_Z}^2 - {k_X}^2} = 1.0$$

Slope of assumed pitching-moment curve for the complete airplane, $\dfrac{-C_m}{\alpha - 20^\circ} = 0.0020$

Lift coefficient $C_L = C_{X''}$ from test data

Each of the parameters was varied, one at a time, from the mean value while all others were kept at the mean value with the exception of C_L, which was equal to $C_{X''}$ for all cases. The values of the parameters chosen are:

μ = 2.5, 5.0, 7.5, and 10.0

$\dfrac{b^2}{{k_Z}^2 - {k_X}^2}$ = 60, 80, 100, and 120

$\dfrac{{k_Z}^2 - {k_Y}^2}{{k_Z}^2 - {k_X}^2}$ = 0.5, 1.0, 1.5, and 2.0

$\dfrac{-C_m}{\alpha - 20^\circ}$ = 0.0010, 0.0015, 0.0020, 0.0025, and 0.0030

The variations in μ include the range of wing loadings of conventional biplanes.

The variations in $\dfrac{b^2}{{k_Z}^2 - {k_X}^2}$ and $\dfrac{{k_Z}^2 - {k_Y}^2}{{k_Z}^2 - {k_X}^2}$ cover the range given in reference 6 for 11 airplanes. These parameters may be written $\dfrac{W\,b^2}{g(C - A)}$ and $\dfrac{C - B}{C - A}$, respectively, where

$A = mk_X{}^2$, the moment of inertia about the X axis

$B = mk_Y{}^2$, the moment of inertia about the Y axis.

$C = mk_Z{}^2$, the moment of inertia about the Z axis.

Method of analysis.- The value of $\Omega b/2V$ was computed for each angle of attack using equation (1). The aerodynamic rolling-moment coefficient required for spinning equilibrium was computed for all values of α and β tested using equation (2). The values of C_m and μ were those used in equation (1). In order to obtain values of C_L ($C_L = C_{X''}$), values of $C_{X''}$, determined from the tests, were plotted against $\Omega b/2V$ and, by interpolation, values of $C_{X''}$ at the values of $\Omega b/2V$ computed from equation (1) were found. By means of similar interpolation, values of C_l were obtained; a correction of 0.02 was added to C_l to give C_l available. The values of C_l available and of C_l required, as found by the preceding methods, were plotted against β, the points of intersection of the two sets of curves giving values of C_l and β, for each angle of attack, at which all forces and moments except yawing moments are in equilibrium.

Values of C_n required to balance the inertia yawing moments were calculated from equation (3), using for C_l the value found for the equilibrium condition. The value of C_n furnished by the wings was the C_n of the tests corrected by adding 0.006. By the subtraction of this value of C_n from the value of C_n required as found by equation (3), the value of C_n was found that must be furnished by the remaining parts of the airplane, fuselage, empennage, and interference effects, to give equilibrium in a steady spin at the given angles of attack.

Scale-effect corrections to C_l ($\Delta C_l = 0.02$) and to C_n ($\Delta C_n = 0.006$) have been found necessary from comparisons of model with full-scale data and are discussed in reference 5.

Discussion of results of analysis.- The angles of sideslip required for a balance of rolling moments and the values of C_n that must be supplied by parts of the air-

plane to balance the inertia couples and wing yawing moments are plotted against the parameters in figures 7 to 14. The results for the 0.25 stagger are included for comparison.

The effect of the various parameters on the sideslip required for equilibrium of rolling moments depends on the angle of attack and on the amount of stagger (figs. 7 to 10). For 50° angle of attack and below, the sideslip is always inward and, except for two cases, is never less than 6°, generally increasing as the stagger decreases (changes in a direction from positive to negative). For an angle of attack of 70° and, sometimes, of 60°, β decreases as the stagger decreases, and for some parameters the sideslip may become outward.

The effect of stagger on C_n required is small. (See figs. 11 to 14.) The C_n required tends to change in the direction from positive toward negative as the stagger increases. The variation of C_n required with the parameters $\frac{-C_m}{\alpha - 20^\circ}$, μ, and $\frac{b^2}{k_Z^2 - k_X^2}$ is usually small, the maximum negative value of C_n required being less than -0.016. The C_n required decreases as $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ increases, the extreme values being 0.013 and -0.023.

<u>Prediction of spinning characteristics of an airplane from the analysis</u>.- Prediction of the spinning characteristics of an airplane in which any of these biplane combinations is used largely depends upon the aerodynamic yawing-moment characteristics of the particular airplane. The value of C_n required, as given in this report, is numerically equal and of opposite sign to the sum of the wing yawing moments and the inertia couples. At any angle of attack, when this value of C_n is supplied by the empennage, fuselage, and interference effects, a steady spin will result provided that the equilibrium is stable; for any other value of C_n the airplane will not spin at that attitude. In order to insure against spinning in any attitude, a value of C_n opposing the spin must be provided that is larger than any attainable value of C_n required. The yawing moment supplied by the empennage, fuselage, and

interference effects depends upon the sideslip; the size and shape of the fuselage and tail surfaces; the location of the horizontal tail surfaces with respect to the fuselage, fin, and rudder; the amount of fin area ahead of the center of gravity; the interference effects between the wings and the rest of the airplane; and the limits of control movements. Data on some of these effects are reported in reference 5 and in references 7 to 12. The geometry of the spin indicates that the vertical tail surfaces should become more effective in producing a yawing moment opposing the spin as the rate of rotation increases and the sideslip decreases. Fin area ahead of the center of gravity will give yawing moments opposing the spin if the sideslip is inward. (See reference 11, fig. 2.)

If the effects of sideslip on the yawing moment supplied by the fuselage, empennage, and interference effects are not considered, for values of stagger tested a biplane with negative stagger will generally have a slightly smaller yawing moment than one with positive stagger. When $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} < 1$ (weight of the airplane distributed along the fuselage, $A < B$), the C_n required opposing the spin will be smallest. When $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} > 1$ (weight of the airplane distributed along the wings, $A > B$), the C_n required opposing the spin will be large and the airplane may be expected to spin flat and recoveries will probably be more difficult.

The effects of sideslip on the yawing moments produced by the tail and fin area indicate that, with large inward sideslip, the vertical tail surfaces would be very ineffective and large amounts of fin area ahead of the wing would be beneficial. In some cases the inward sideslip at the center of gravity may be large enough to make the sideslip at the tail inward, in which case the tail and the fuselage behind the center of gravity would furnish yawing moments aiding the spin. It follows that two general methods of preventing a dangerous spin might be considered.

The first method is to design an airplane that will attain spinning equilibrium with as small an amount of inward sideslip as possible so that the rear part of the

fuselage and the tail surfaces will have maximum effectiveness. A tail with a large unshielded vertical fin area will then give the maximum obtainable yawing moment opposing the spin. A large diving moment, a small value of $\frac{b^2}{k_Z^2 - k_X^2}$, a large value of wing loading, and a large value of $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ are factors giving the smallest amounts of inward sideslip, although the large values of $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$ also give relatively large values of C_n required opposing the spin.

The second method is based on the assumption that an appreciable yawing moment opposing the spin may be set up by fin area ahead of the center of gravity (reference 11). This yawing moment would be expected to increase as the inward sideslip and the vertical fin area ahead of the center of gravity increase. The airplane should then be designed with the maximum possible vertical fin area ahead of the center of gravity; and, to obtain maximum inward sideslip, a small diving moment, a large value of $\frac{b^2}{k_Z^2 - k_X^2}$, lightly loaded wings, and a small value of $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$.

A good tail arrangement, i.e., one with a large unshielded fin area, may not always prevent flat spins because, for some cases, the sideslip at the tail may be zero or inward, which will result in a tail yawing moment of zero or even aiding the spin.

CONCLUSIONS

On the assumption that the arbitrary constants added to the rolling-moment and yawing-moment coefficients are

of the right order of magnitude, the following conclusions are indicated by the analysis presented for a conventional biplane with rectangular Clark Y wings having 0.25, zero, and -0.25 stagger, gap equal to the chord, and 0° decalage:

1. The value of the yawing-moment coefficient required from the fuselage, tail, and interference effects for steady spinning equilibrium at any angle of attack is small and nearly always negative (opposing the rotation) throughout the range investigated.

2. The maximum value of the yawing-moment coefficient that must be supplied by all parts of the airplane other than the wings and inertia couples to prevent spinning equilibrium at any angle of attack is $C_n = -0.025$.

3. The value of stagger for the best spinning characteristics varies with different types of airplanes.

4. At some angles of attack, the inward sideslip will be very great (more than 20°) so that even good tail arrangements may have little effect in preventing a dangerous spin; fin area ahead of the wings will be beneficial.

5. The angle of attack at which the maximum inward sideslip occurs decreases as the stagger changes from positive toward negative. For angles of attack through 50°, the sideslip generally becomes more inward as the stagger becomes more negative, the opposite being true at 70° angle of attack, with the transition taking place at some intermediate angle of attack.

6. Too much reliance should not be placed on tail arrangement for preventing bad spinning characteristics.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., October 19, 1937.

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TABLE I. Aerodynamic Characteristics of a Clark Y Biplane Cellule - 0.25 stagger

$\beta = 0^\circ$

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
0.25	70	0.280	0.052	-0.002	-0.677	0.0095	0.0048	-0.0013
.50	70	.297	.022	-.001	-.807	.0192	.0045	-.0030
.75	70	.364	-.002	-.002	-1.070	.0154	-.0143	-.0028
1.00	70	.533	.022	.010	-1.499	-.0362	-.0450	-.0010

$\beta = 5^\circ$

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
0.25	30	0.777	-0.028	-0.005	-0.904	0.0008	0.0028	0.0038
	40	.688	-.023	-.009	-.916	.0046	.0055	-.0019
	50	.516	.000	-.008	-.802	.0252	.0095	-.0042
	60	.397	.042	-.002	-.722	.0136	.0093	-.0046
	70	.289	.062	-.001	-.675	-.0061	.0068	-.0013
.50	30	.822	-.025	.005	-.964	.0062	.0062	.0054
	40	.633	-.057	-.009	-.874	.0270	.0057	-.0025
	50	.529	-.029	-.011	-.857	.0321	.0062	-.0067
	60	.420	.000	-.009	-.840	.0298	.0077	-.0059
	70	.290	.022	-.001	-.788	.0146	.0067	-.0031
.75	30	.884	-.031	.017	-1.039	-.0404	.0026	.0005
	40	.800	-.052	.003	-1.088	.0235	.0037	.0013
	50	.621	-.045	-.005	-1.020	.0176	.0025	-.0039
	60	.509	-.020	-.010	-1.053	.0272	.0000	-.0056
	70	.365	.008	-.001	-1.044	.0122	-.0021	-.0020
1.00	30	.986	-.019	.032	-1.150	-.1290	-.0006	-.0042
	40	.974	-.049	.024	-1.313	-.0313	-.0055	.0016
	50	.819	-.044	.012	-1.325	-.0118	-.0036	.0007
	60	.637	-.034	.002	-1.333	-.0103	-.0117	-.0023
	70	.524	.023	-.004	-1.468	-.0261	-.0231	.0008

$\beta = 10^\circ$

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
0.25	30	0.757	-0.031	-0.005	-0.893	-.0164	0.0023	0.0032
	40	.655	-.036	-.005	-.886	-.0161	.0051	-.0014
	50	.506	-.012	-.001	-.801	.0053	.0106	-.0032
	60	.369	.021	.002	-.703	.0040	.0100	-.0050
	70	.284	.057	.003	-.672	-.0165	.0054	-.0010
.50	30	.774	-.038	.002	-.916	-.0009	.0057	.0050
	40	.628	-.043	-.008	-.856	.0164	.0055	-.0020
	50	.490	-.045	-.012	-.816	.0269	.0062	-.0065
	60	.402	-.009	-.001	-.819	.0268	.0073	-.0059
	70	.284	.024	.001	-.766	.0104	.0065	-.0030
.75	30	.870	-.027	.012	-1.020	-.0337	.0020	.0001
	40	.772	-.046	-.001	-1.046	.0196	.0041	-.0009
	50	.624	-.035	-.008	-1.012	.0131	.0014	-.0049
	60	.489	-.022	-.013	-1.016	.0208	.0040	-.0060
	70	.331	-.015	-.005	-1.008	.0140	.0008	-.0022
1.00	30	.989	-.021	.029	-1.153	-.1132	-.0027	-.0023
	40	.987	-.029	.023	-1.312	-.0325	-.0034	-.0005
	50	.802	-.033	.006	-1.287	-.0104	-.0051	-.0010
	60	.618	-.033	-.006	-1.293	-.0054	-.0068	-.0014
	70	.471	-.016	-.004	-1.420	-.0116	-.0177	-.0001

$\beta = 15^\circ$

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
0.25	30	0.737	-0.050	-0.001	-0.880	-0.0342	0.0005	0.0029
	40	.655	-.038	.000	-.887	-.0333	.0035	-.0018
	50	.500	-.024	.001	-.807	-.0159	.0085	-.0020
	60	.316	-.026	.001	-.677	-.0081	.0092	-.0037
	70	.200	-.020	.005	-.642	-.0230	.0049	-.0010
.50	30	.778	-.033	.000	-.917	-.0117	.0046	.0043
	40	.660	-.015	-.009	-.874	.0041	.0056	-.0008
	50	.502	-.023	-.009	-.810	.0233	.0069	-.0065
	60	.402	.001	-.002	-.801	.0282	.0090	-.0064
	70	.293	.037	.000	-.754	.0056	.0063	-.0032
.75	30	.840	-.034	.012	-.989	-.0271	-.0051	.0007
	40	.750	-.018	.003	-.995	.0142	-.0054	-.0007
	50	.647	.013	-.002	-.991	.0134	-.0062	-.0041
	60	.521	.022	-.007	-1.004	.0242	-.0033	-.0047
	70	.388	.043	-.002	-1.018	.0183	-.0025	-.0009
1.00	30	1.030	.001	.031	-1.189	-.1029	-.0175	-.0014
	40	1.012	.018	.024	-1.306	-.0254	-.0186	.0009
	50	.845	.037	.009	-1.270	-.0099	-.0161	-.0009
	60	.686	.049	.010	-1.285	.0014	-.0179	.0000
	70	.558	.077	.001	-1.420	.0058	-.0157	.0006

$\beta = 20^\circ$

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
0.25	30	0.792	-0.024	0.000	-0.928	-0.0465	-.0008	0.0032
	40	.697	-.011	.005	-.919	-.0476	.0023	-.0010
	50	.452	-.069	.005	-.785	-.0339	-.0059	.0017
	60	.397	.037	.004	-.730	-.0252	.0074	-.0028
	70	.299	.063	.006	-.700	-.0389	.0037	-.0010
.50	30	.739	-.045	.003	-.880	-.0262	-.0002	.0044
	40	.627	-.036	-.006	-.849	-.0121	.0008	.0007
	50	.487	-.035	-.008	-.799	.0156	.0044	-.0048
	60	.391	-.004	-.003	-.790	.0223	.0065	-.0051
	70	.261	.006	.000	-.744	-.0035	.0041	-.0023
.75	30	.779	-.051	.008	-.930	-.0304	-.0016	.0002
	40	.701	-.027	.000	-.938	.0069	.0046	-.0023
	50	.581	-.016	-.003	-.923	.0145	-.0012	-.0038
	60	.497	.012	-.007	-.972	.0315	.0043	-.0062
	70	.350	.011	-.004	-.994	.0204	.0024	-.0021
1.00	30	.880	-.057	.023	-1.049	-.0883	-.0114	-.0018
	40	.862	-.034	.015	-1.153	-.0248	-.0013	-.0030
	50	.775	.016	.011	-1.186	-.0083	-.0118	-.0003
	60	.619	.012	.000	-1.341	.0057	-.0036	-.0031
	70	.467	.004	.002	-1.354	.0160	-.0073	-.0013

Coefficients of forces and moments given for and about the quarter-chord point at the lower surface of the upper wing.

TABLE II. Aerodynamic Characteristics of a Clark Y Biplane Cellule
Zero stagger

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
$\beta = 0°$								
0.25	30	0.833	-0.042	-0.007	-0.986	0.0158	-0.0228	0.0061
	40	.753	-.040	-.012	-1.017	.0135	-.0270	-.0008
	50	.611	-.030	-.009	-.986	.0205	-.0248	-.0014
	60	.436	-.010	-.002	-.889	.0357	-.0248	-.0017
	70	.294	.031	-.003	-.774	.0312	-.0258	-.0035
.50	30	.917	-.033	.007	-1.078	-.0080	-.0260	.0054
	40	.813	-.034	-.006	-1.090	.0222	-.0315	.0033
	50	.629	-.043	-.016	-1.030	.0493	-.0305	-.0011
	60	.449	-.034	-.011	-.958	.0507	-.0300	-.0045
	70	.304	-.009	-.004	-.915	.0473	-.0323	-.0042
.75	30	1.017	-.031	.023	-1.193	-.0629	-.0300	.0026
	40	.981	-.022	.011	-1.299	.0177	-.0390	.0044
	50	.776	-.017	-.005	-1.227	.0468	-.0453	.0011
	60	.564	-.027	-.012	-1.174	.0499	-.0458	-.0027
	70	.380	-.014	-.012	-1.150	.0559	-.0450	-.0037
1.00	30	1.111	-.027	.043	-1.299	-.1576	-.0323	-.0006
	40	1.146	-.001	.036	-1.497	-.0541	-.0530	.0064
	50	.917	-.030	.023	-1.461	.0109	-.0678	.0067
	60	.641	-.067	.002	-1.398	.0257	-.0603	.0019
	70	.475	-.025	-.007	-1.455	.0397	-.0650	-.0002
$\beta = 5°$								
0.25	30	0.823	-0.038	-0.008	-0.973	0.0013	-0.0237	0.0055
	40	.746	-.039	-.012	-1.007	-.0036	-.0272	-.0013
	50	.622	-.023	-.007	-.995	.0040	-.0253	-.0017
	60	.448	-.001	.000	-.898	.0225	-.0237	-.0014
	70	.265	.005	.000	-.761	.0216	-.0255	-.0031
.50	30	.895	-.037	.002	-1.055	-.0131	-.0259	.0047
	40	.792	-.035	-.008	-1.063	.0101	-.0301	.0023
	50	.624	-.029	-.015	-1.006	.0339	-.0296	-.0017
	60	.463	-.019	-.011	-.960	.0436	-.0308	-.0040
	70	.296	-.016	-.003	-.910	.0439	-.0310	-.0039
.75	30	1.026	-.029	.021	-1.201	-.0592	-.0343	.0019
	40	.976	-.011	.006	-1.283	.0146	-.0418	.0031
	50	.750	-.010	-.006	-1.179	.0362	-.0429	.0006
	60	.607	.018	-.012	-1.184	.0470	-.0436	-.0023
	70	.374	-.014	-.012	-1.131	.0519	-.0424	-.0038
1.00	30	1.154	-.016	.038	-1.341	-.1439	-.0441	.0001
	40	1.158	.007	.029	-1.505	-.0439	-.0589	.0044
	50	.944	.001	.016	-1.469	.0045	-.0636	.0037
	60	.754	.043	.004	-1.432	.0239	-.0619	.0015
	70	.462	-.025	-.012	-1.420	.0452	-.0594	.0003

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
$\beta = 10°$								
0.25	30	0.809	-0.041	-0.005	-0.958	-0.0154	-0.0241	0.0052
	40	.737	-.035	-.010	-.992	-.0232	-.0274	-.0015
	50	.637	-.013	.001	-1.006	-.0145	-.0270	-.0013
	60	.448	-.009	.004	-.912	.0064	-.0244	-.0011
	70	.274	.012	.003	-.766	.0124	-.0256	-.0028
.50	30	.874	-.040	.002	-1.032	-.0181	-.0282	.0045
	40	.774	-.026	-.008	-1.032	-.0009	-.0286	.0027
	50	.618	-.016	-.011	-.980	.0196	-.0283	-.0021
	60	.480	.001	-.006	-.959	.0371	-.0321	-.0036
	70	.304	-.003	.000	-.897	.0426	-.0311	-.0042
.75	30	.998	-.039	.016	-1.175	-.0582	-.0329	.0010
	40	.905	-.034	.004	-1.210	.0050	-.0405	.0020
	50	.712	-.018	-.008	-1.130	.0276	-.0440	.0008
	60	.562	-.006	-.013	-1.135	.0390	-.0460	-.0024
	70	.380	-.003	-.007	-1.120	.0497	-.0446	-.0033
1.00	30	1.141	-.029	.033	-1.334	-.1339	-.0426	.0000
	40	1.096	-.040	.025	-1.464	-.0354	-.0620	.0035
	50	.905	-.009	.007	-1.419	.0051	-.0662	.0042
	60	.707	.009	-.005	-1.398	.0232	-.0619	.0019
	70	.473	-.011	-.006	-1.413	.0505	-.0665	.0020
$\beta = 15°$								
0.25	30	0.816	-0.037	-0.003	-0.964	-0.0322	-0.0244	0.0050
	40	.746	-.029	-.002	-.998	-.0388	-.0279	-.0010
	50	.637	-.021	.007	-1.016	-.0355	-.0274	-.0015
	60	.457	-.009	.004	-.929	-.0122	-.0254	-.0006
	70	.251	-.009	.001	-.760	-.0002	-.0263	-.0022
.50	30	.842	-.050	.001	-1.001	-.0255	-.0284	.0043
	40	.747	-.032	-.008	-1.002	-.0157	-.0296	.0019
	50	.610	-.023	-.007	-.975	.0068	-.0309	-.0014
	60	.460	-.006	-.001	-.929	.0315	-.0316	-.0028
	70	.306	.006	-.003	-.878	.0407	-.0319	-.0038
.75	30	.939	-.052	.015	-1.115	-.0565	-.0334	.0010
	40	.866	-.030	.000	-1.155	-.0031	-.0390	.0015
	50	.708	.002	-.007	-1.099	.0209	-.0393	.0003
	60	.536	-.008	-.011	-1.086	.0343	-.0390	-.0038
	70	.384	.006	-.005	-1.106	.0544	-.0409	-.0033
1.00	30	1.110	-.054	.030	-1.313	-.1204	-.0459	.0008
	40	1.084	-.007	.019	-1.421	-.0303	-.0551	.0017
	50	.866	-.001	.002	-1.348	.0029	-.0595	.0020
	60	.740	.052	-.007	-1.390	.0227	-.0597	.0009
	70	.481	.006	-.009	-1.391	.0518	-.0562	-.0001

$\frac{\Omega b}{2V}$	α (deg.)	$C_{X''}$	C_X	C_Y	C_Z	C_l	C_m	C_n
$\beta = -5°$								
0.25	30	0.875	-0.057	-0.008	-1.044	0.0300	-0.0239	0.0056
	40	.806	-.016	-.015	-1.065	.0282	-.0269	-.0006
	50	.669	.010	-.011	-1.029	.0345	-.0252	-.0017
	60	.475	.024	-.008	-.810	.0518	-.0245	-.0022
	70	.302	.032	-.007	-.795	.0445	-.0259	-.0034
.50	30	.972	-.038	.007	-1.145	.0015	-.0270	.0066
	40	.894	-.005	-.004	-1.171	.0363	-.0286	.0053
	50	.681	-.023	-.015	-1.088	.0625	-.0330	-.0019
	60	.477	-.015	-.010	-.980	.0612	-.0319	-.0050
	70	.318	-.003	-.007	-.938	.0552	-.0313	-.0039
.75	30	1.048	-.088	.026	-1.262	-.0402	-.0368	.0073
	40	.980	-.035	.015	-1.308	.0147	-.0435	.0063
	50	.818	-.011	.002	-1.286	.0531	-.0488	.0020
	60	.610	-.001	-.008	-1.222	.0584	-.0472	-.0035
	70	.416	.013	-.008	-1.182	.0631	-.0437	-.0039
1.00	30	1.207	-.024	.039	-1.408	-.1897	-.0387	.0068
	40	1.204	.026	.044	-1.549	-.0621	-.0557	.0062
	50	.988	.002	.034	-1.535	.0109	-.0730	.0099
	60	.762	.031	.012	-1.471	.0233	-.0700	.0038
	70	.554	.039	.003	-1.513	.0314	-.0663	.0006

Coefficients of forces and moments given for and about the quarter-chord point at the lower surface of the upper wing.

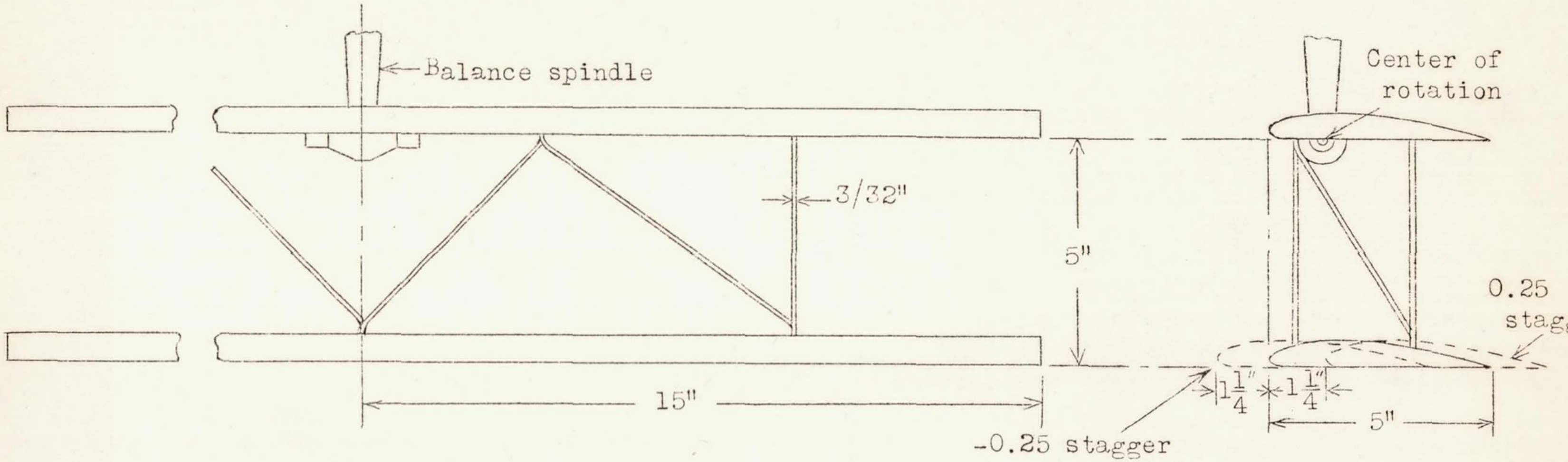


Figure 1.- Clark Y biplane cellule.

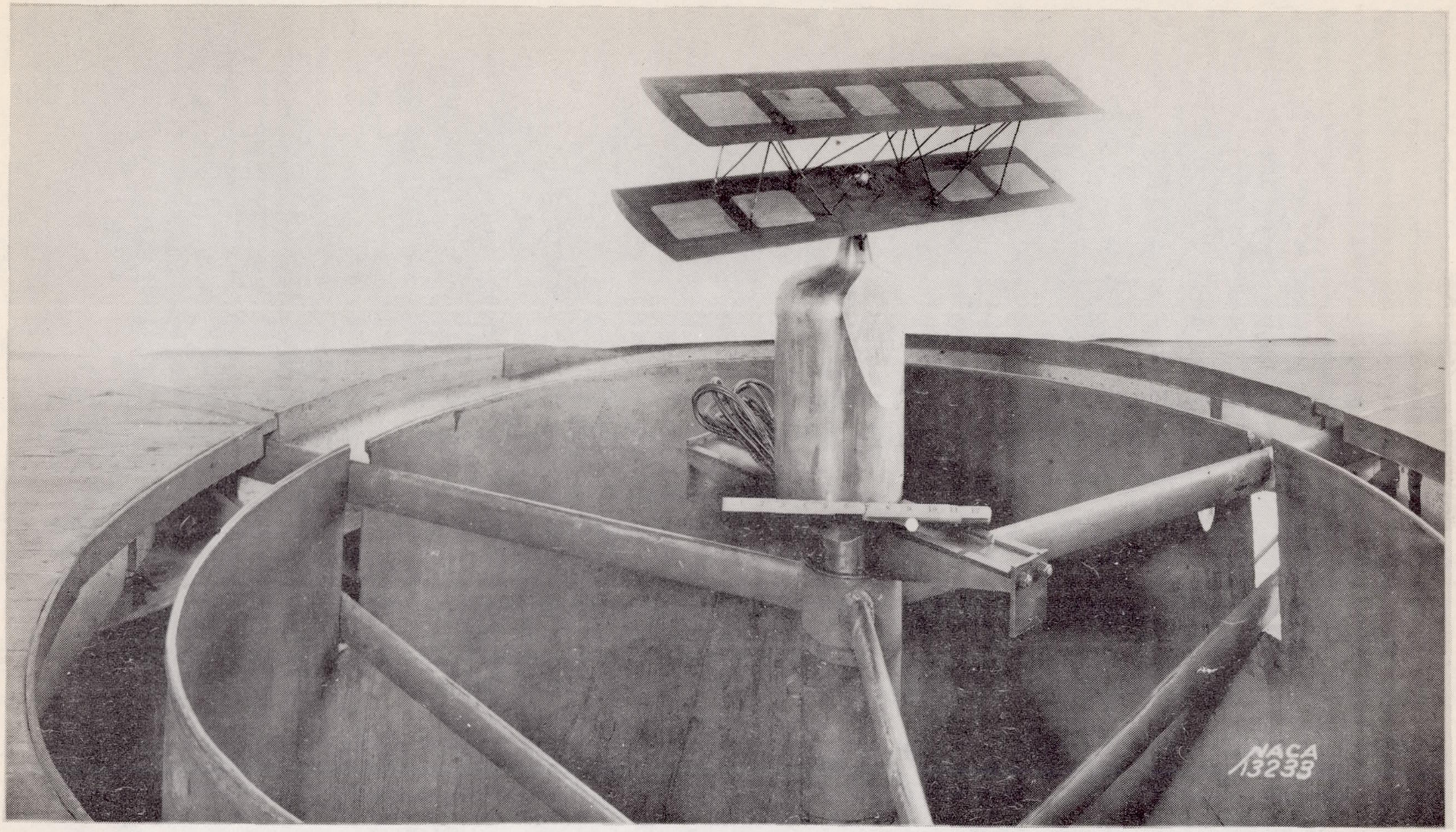


Figure 2.- The rectangular Clark Y biplane cellule, 0.25 stagger, mounted on the spinning balance.

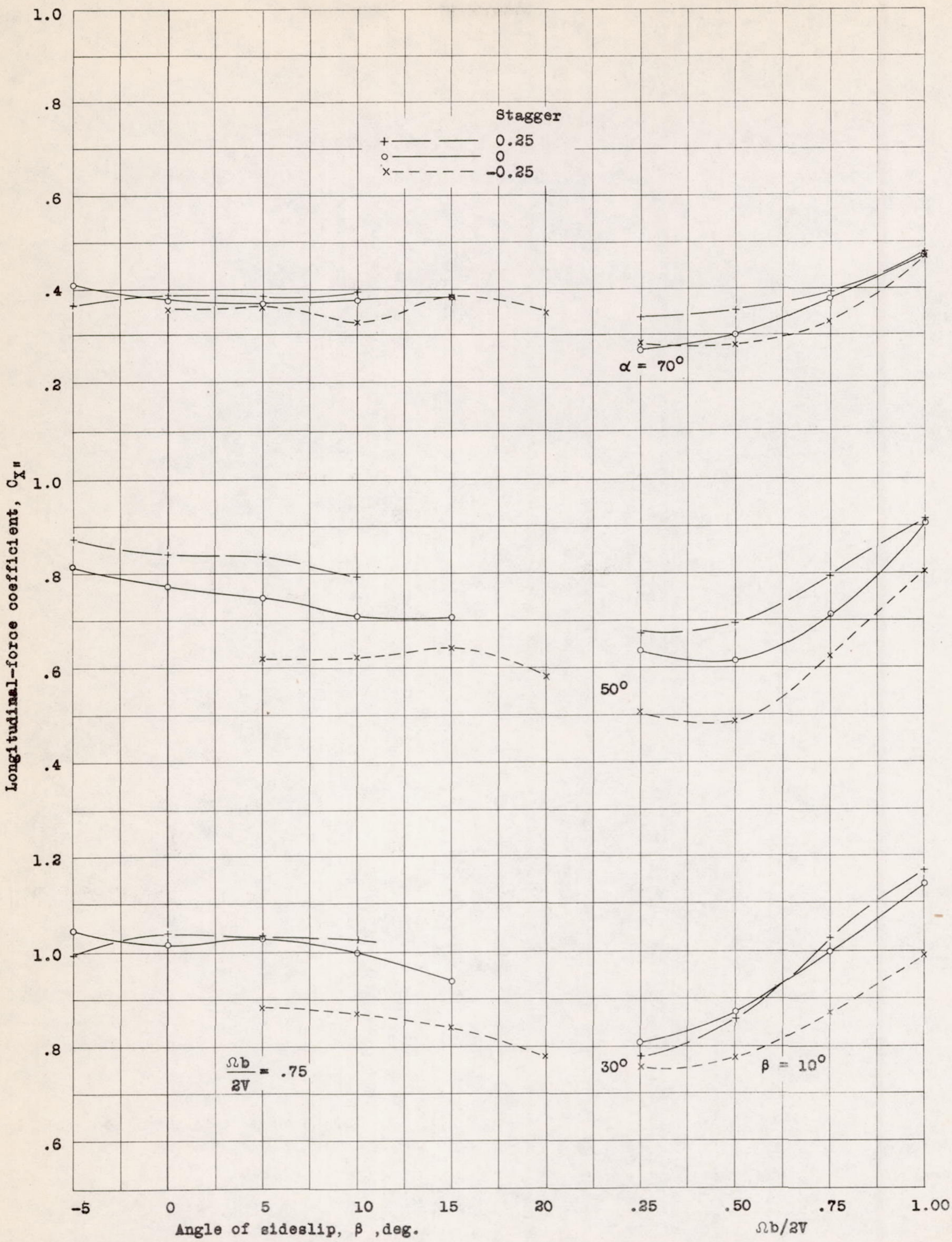


Figure 3.- Variation of longitudinal-force coefficient, $C_{X''}$ (earth axes) with angle of sideslip and with $\Omega b/2V$.

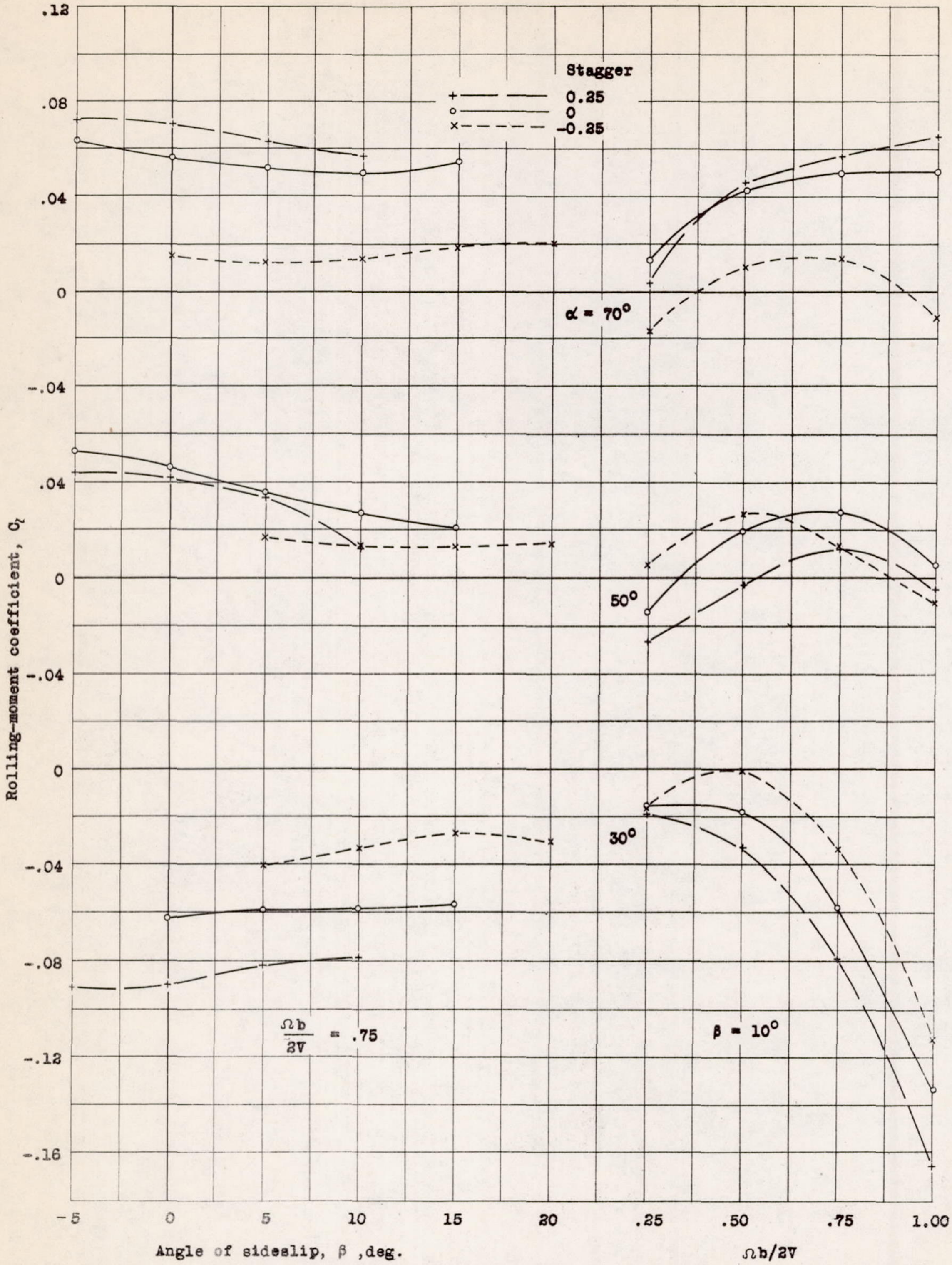


Figure 4.- Variation of rolling-moment coefficient C_l (body axes) with angle of sideslip and $\Omega b/2V$.

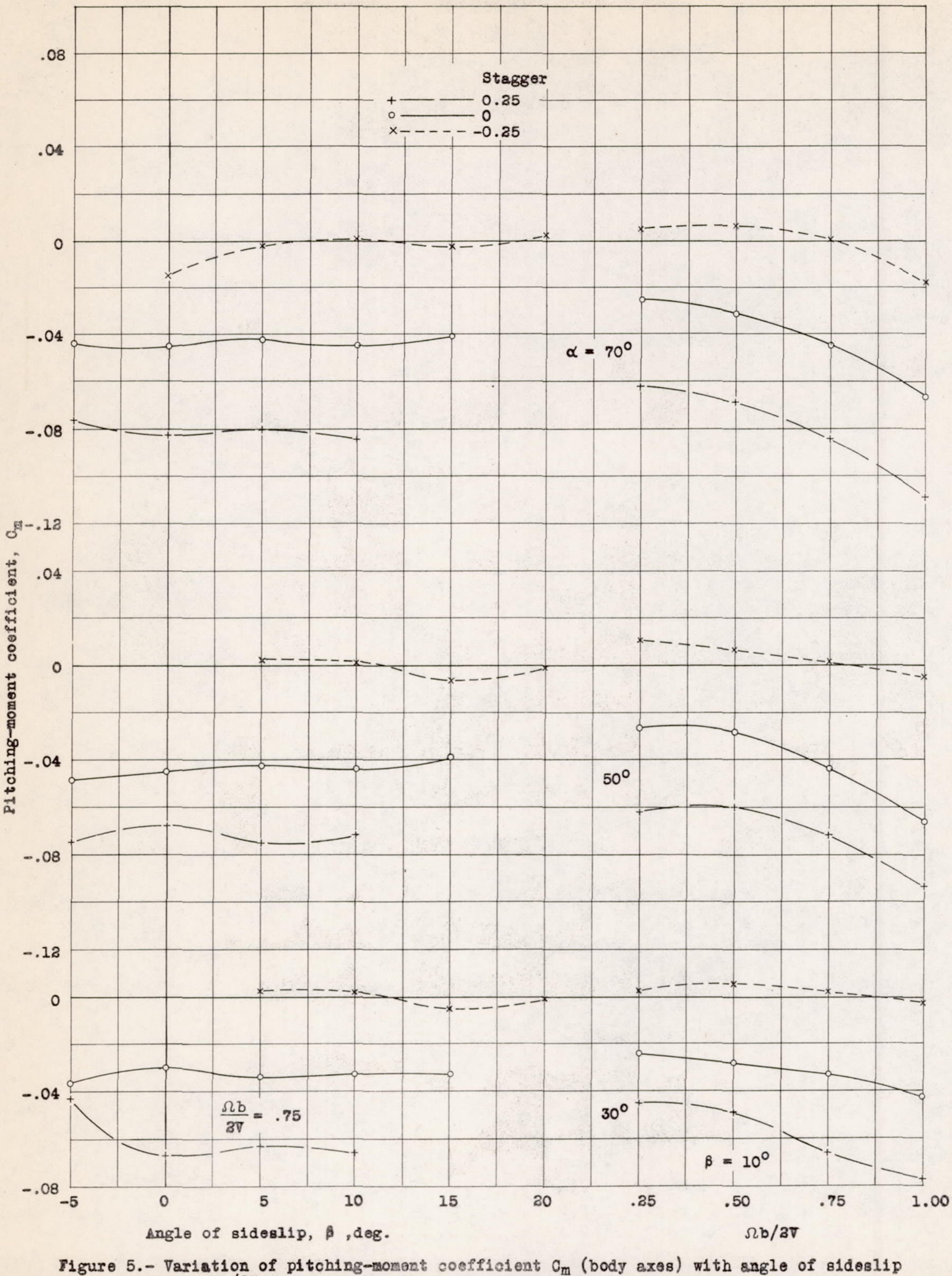


Figure 5.- Variation of pitching-moment coefficient C_m (body axes) with angle of sideslip and $\Omega b/2V$.

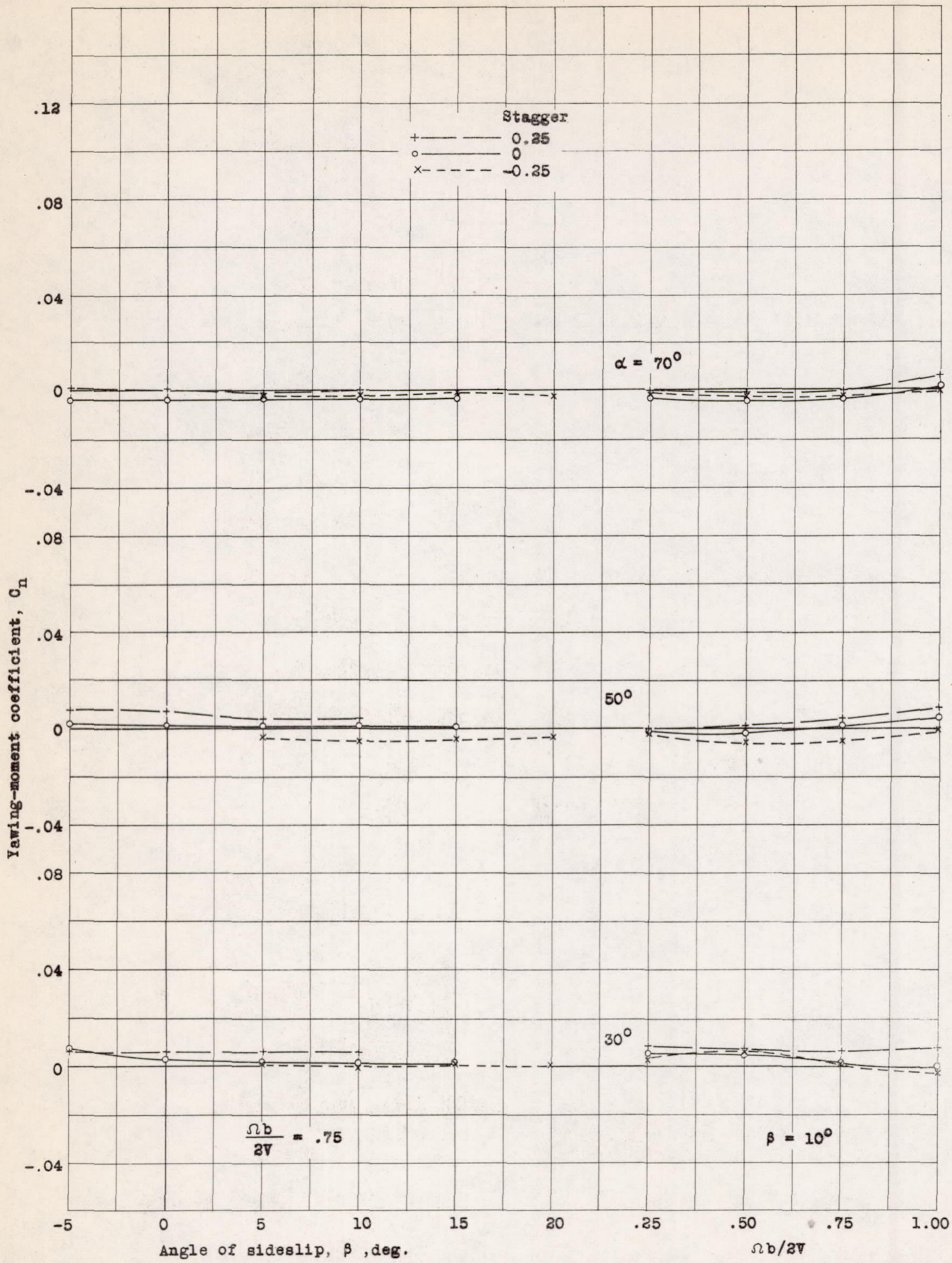


Figure 6.- Variation of yawing-moment coefficient C_n(body axes) with angle of sideslip and with $\Omega b/2V$.

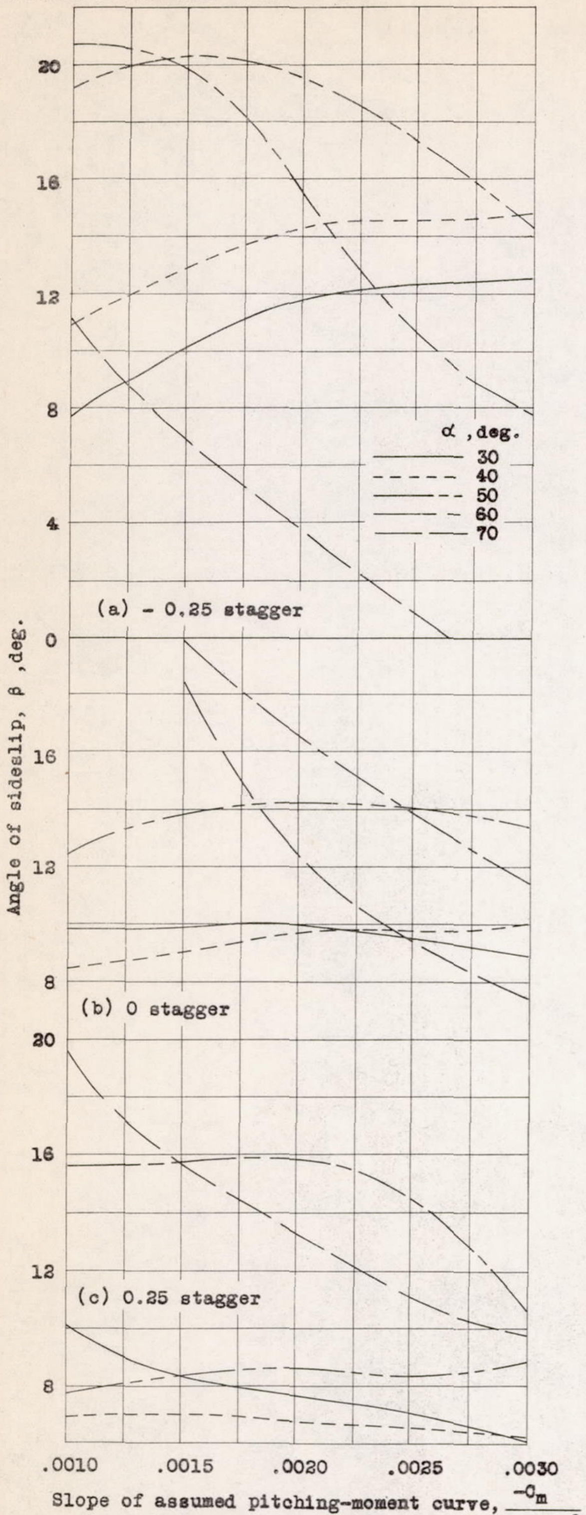


Figure 7.- Effect of pitching-moment coefficient upon sideslip necessary for equilibrium in a spin. $\mu = 5$ $C_L = C_{X''}$

$\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$ $\frac{b^2}{k_Z^2 - k_X^2} = 80$

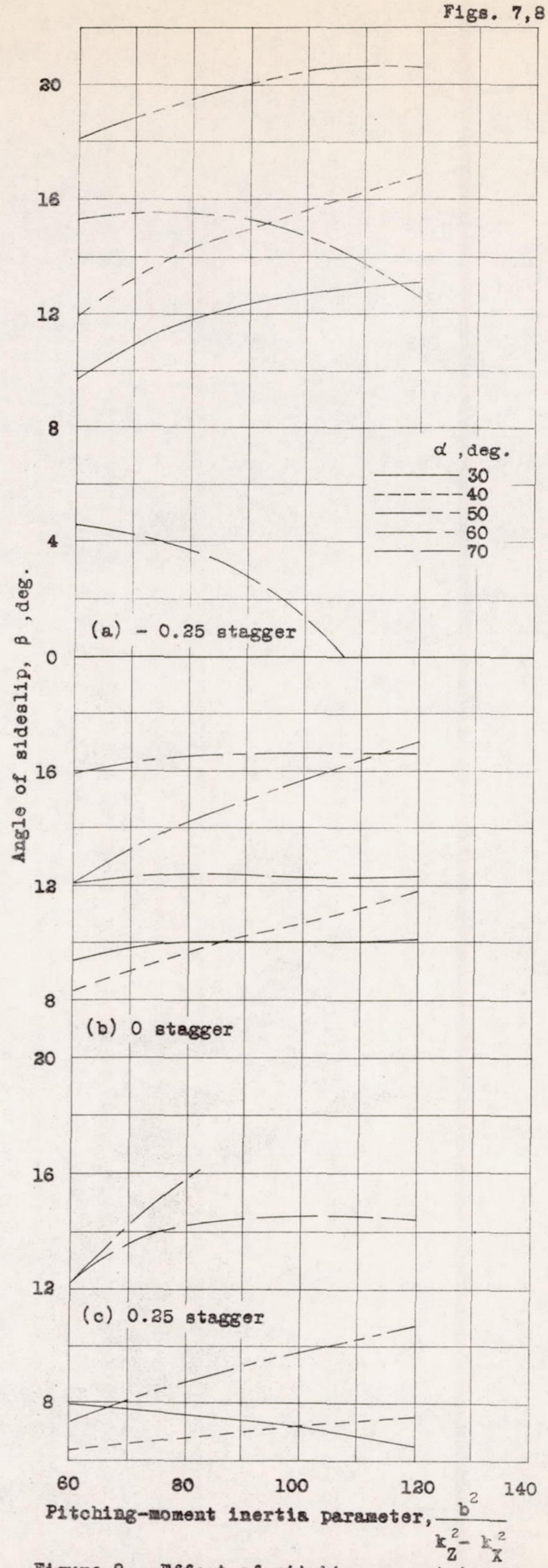


Figure 8.- Effect of pitching-moment inertia parameter upon sideslip necessary for equilibrium in a spin. $\mu = 5$ $C_L = C_{X''}$

$C_m = -0.0020(\alpha - 20^\circ)$, $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$

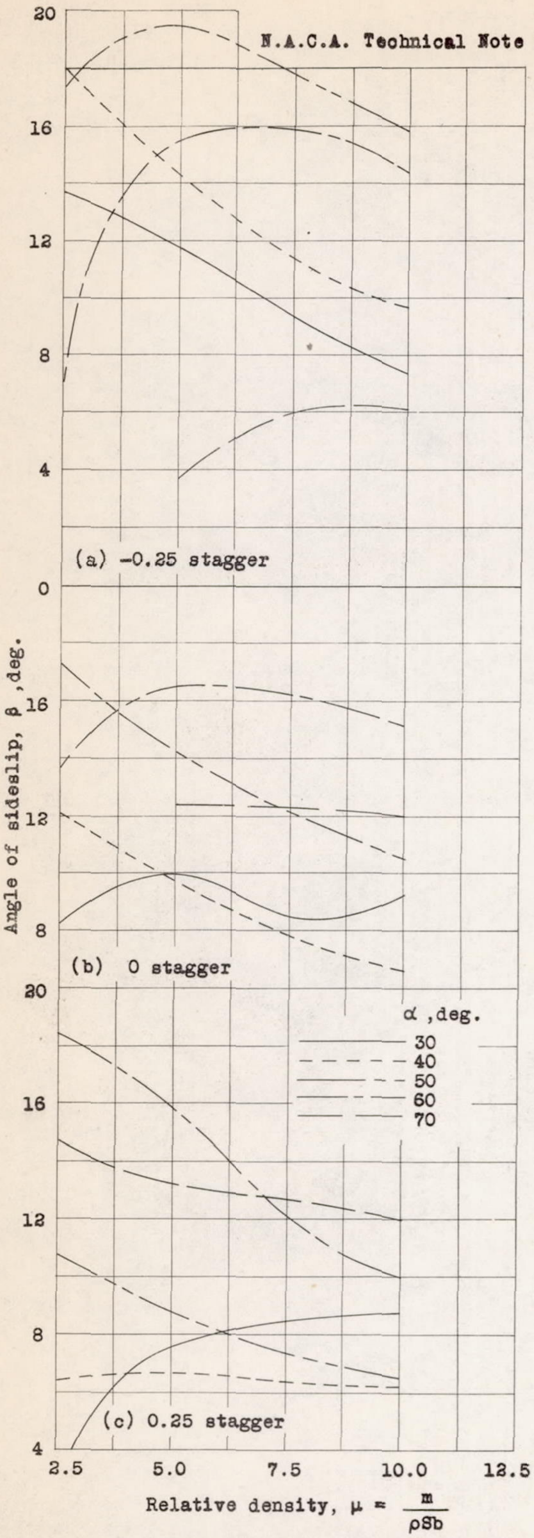


Figure 9.- Effect of relative density of airplane upon sideslip necessary for equilibrium in a spin. $C_m = -0.0020(\alpha - 20^\circ)$

$C_L = C_{X''}$ $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$ $\frac{b^2}{k_Z^2 - k_X^2} = 80$

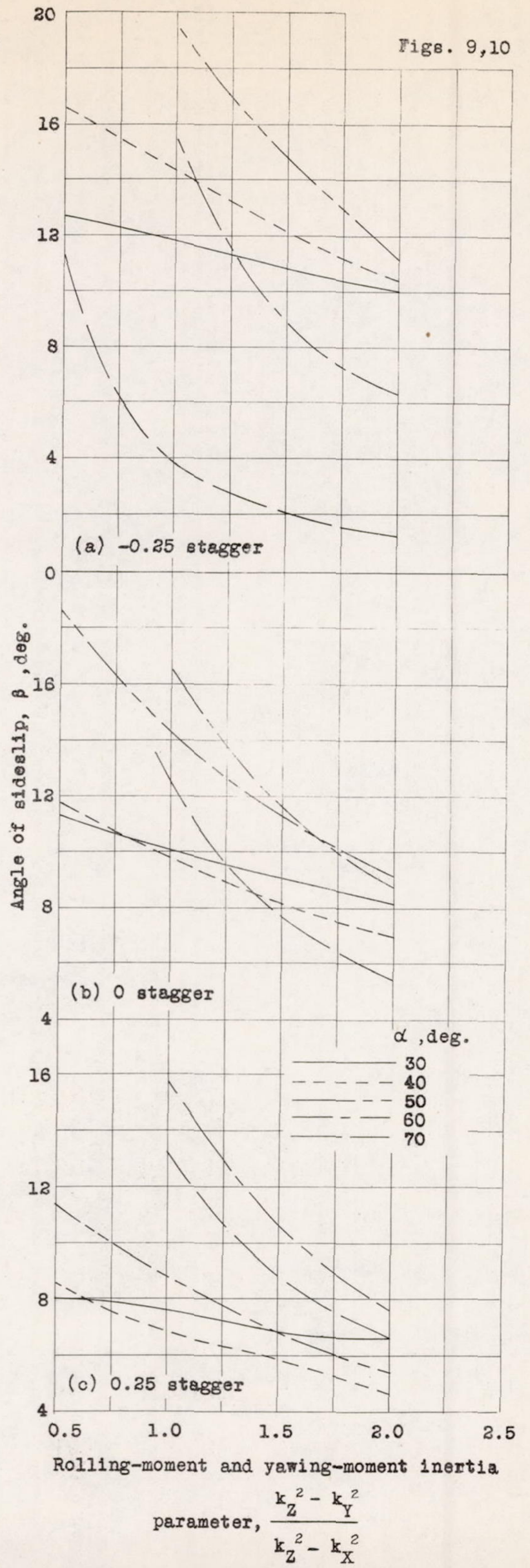


Figure 10.- Effect of rolling-moment and yawing-moment inertia parameter upon sideslip necessary for equilibrium in a spin. $\mu = 5$ $C_L = C_{X''}$

$C_m = -0.0020(\alpha - 20^\circ)$ $\frac{b^2}{k_Z^2 - k_X^2} = 80$

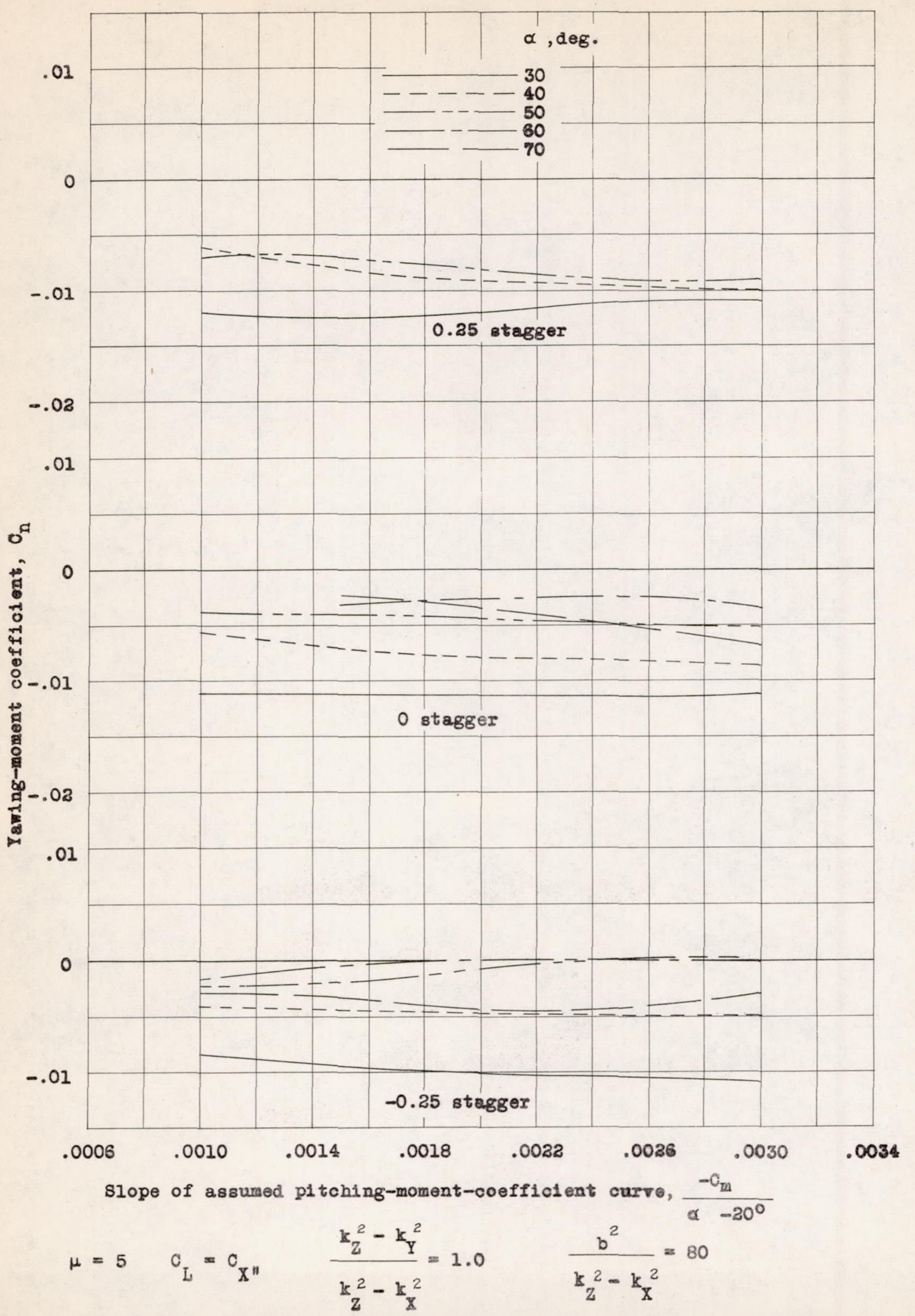
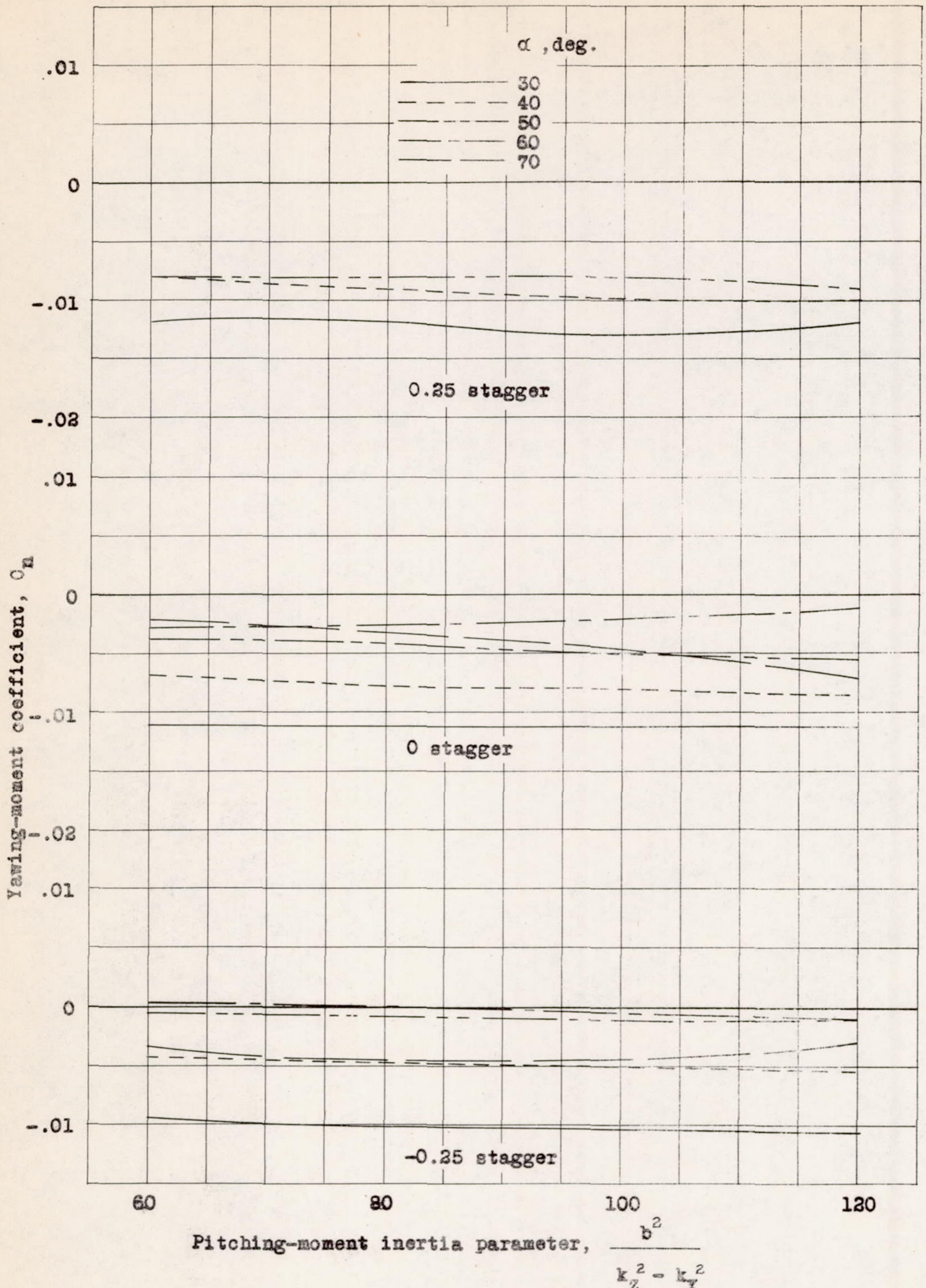
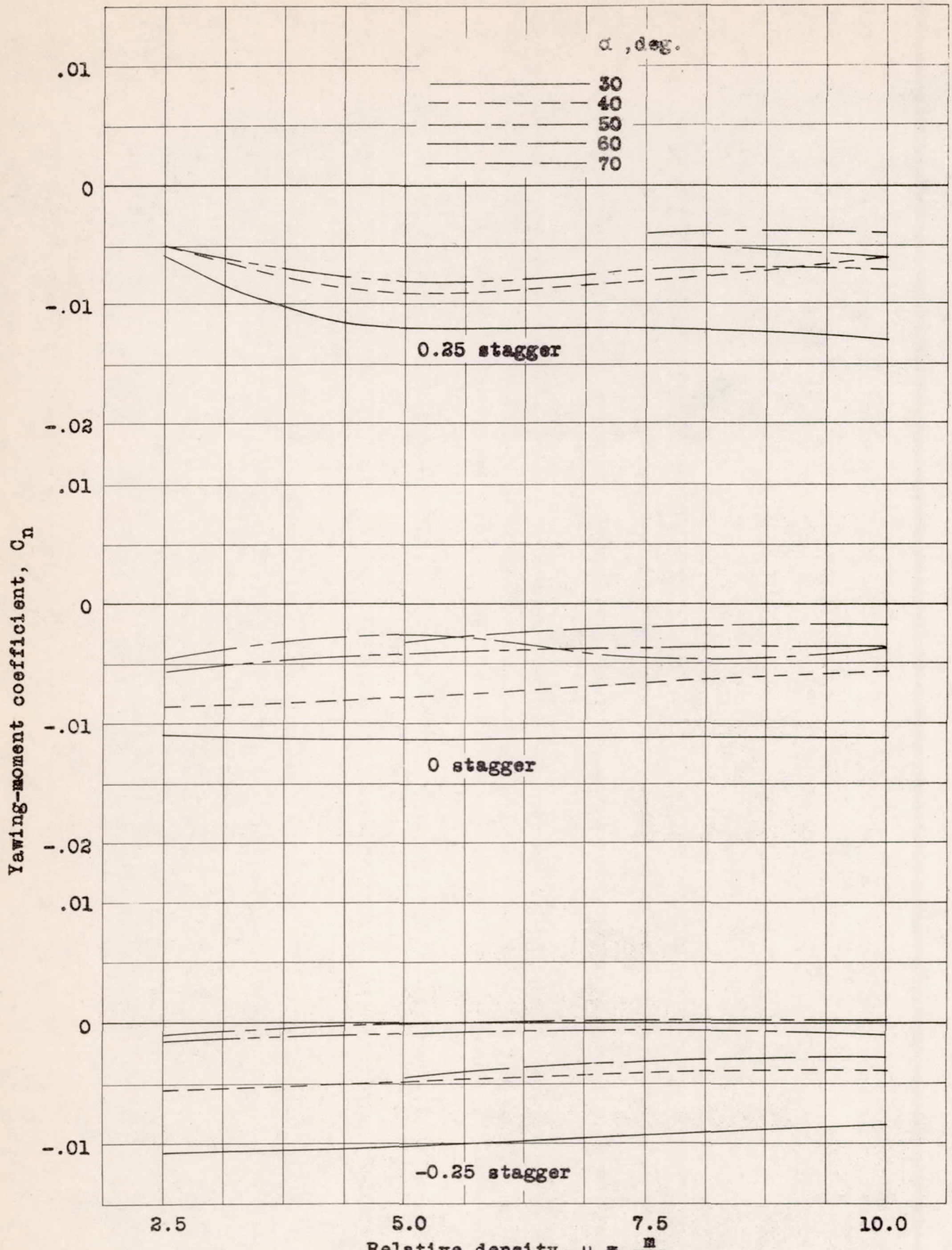


Figure 11.- Effect of pitching-moment coefficient upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin.



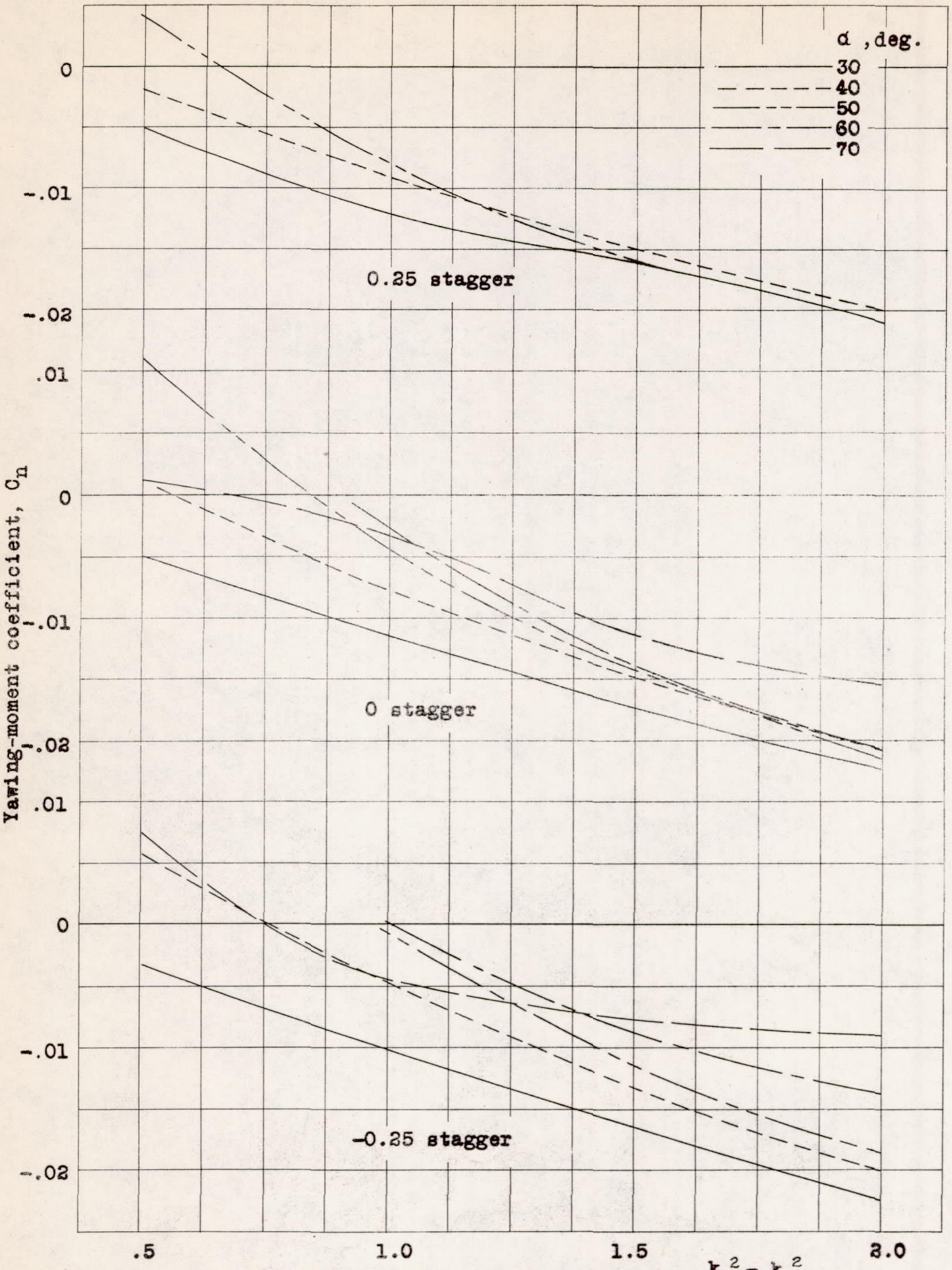
$\mu = 5$ $C_L = C_{X''}$ $C_m = -0.0020(\alpha - 20°)$ $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$

Figure 12.- Effect of pitching-moment inertia parameter upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin.



$C_m = -0.0020(\alpha - 20^\circ)$ $C_L = C_{X''}$ $\frac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2} = 1.0$ $\frac{b^2}{k_Z^2 - k_X^2} = 80$

Figure 13.- Effect of relative density of airplane upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin.



Rolling-moment and yawing-moment inertia parameter, $\dfrac{k_Z^2 - k_Y^2}{k_Z^2 - k_X^2}$

$\mu = 5 \qquad C_L = C_{X''} \qquad C_m = -0.0020(\alpha - 20^\circ) \qquad \dfrac{b^2}{k_Z^2 - k_X^2} = 80$

Figure 14.- Effect of rolling-moment and yawing-moment inertia parameter upon yawing-moment coefficient that must be supplied by parts other than the wing for equilibrium in a spin.